BELLCOMM, INC.
955 L'ENFANT PLAZA NORTH, S.W. WASHINGTON, D. C. 20024

B70 01020

SUBJECT: Radiative Thermal Protection System Considerations for the Space Shuttle - Case 105-6

DATE: January 16, 1970

FROM: C. C. Ong

ABSTRACT

Ablative materials have performed quite satisfactorily as heat shields to protect manned spacecraft from the searing temperatures of entering the earth's atmosphere, but the time and difficulty to replace eroded surfaces are inconvenient and costly for reusable earth-to-space transportation systems. A more desirable thermal protection system (TPS) for the Space Shuttle would be based upon the use of a high temperature resistant material that could reject heat by radiation and be used many times without repair or replacement. Alloys of nickel, columbium, tantalum, and an all silica composite have shown considerable promise for fulfilling this requirement, but they need further technological development and their use may still impose restrictions on vehicle shape and flight path. To insure flight operation at an early date, it would be ideal to have a thermal protection system that could interchangeably use ablative or radiative heat shield materials. The ablative material could serve as an evolutionary step or as a backup to the more desirable radiative heat shield.

A radiative TPS would consist of a metallic heat shield with internal and/or external non-metallic insulation, possibly including ablative sections or circulating liquid to cool the vehicle nose and leading edges, all supported by a separate load bearing structure. A non-metallic composite might also be used to serve both as a heat radiator and as an insulator. It might be possible to use a metallic heat shield as primary load bearing structure, but this can be applied advantageously only for thin structures such as fins and control surfaces. Selection of a TPS configuration depends on a number of interrelated problems, the most critical of which are: (1) the evaluation and selection of heat shield and insulation materials, (2) control of structural weight, (3) control of reentry surface temperatures, (4) determination of the effects of the impact of meteoroid and other foreign objects, and (5) protection of special regions such as doors, booster-orbiter attachment, and turbojet landing engines.

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MEMORANDUM FOR FILE

I. Introduction

The design of a reliable, reusable thermal protection system (TPS) is generally considered one of the most critical technology areas in the development of a reusable space shuttle vehicle. Ablative TPS have so far been used in the very high temperature surface areas of present manned reentry vehicles, and have been quite satisfactory in fulfilling the requirements for a single reentry. Ablative systems, however, might have many shortcomings if applied to a reusable shuttle vehicle. Past studies[1-6] indicate that a radiative TPS would be a more advantageous system for such a reusable reentry vehicle.

Another Bellcomm study[6] includes discussions on the availability, reusability and the prospects for future development of several high temperature materials which appear promising for the heat shield of a radiative TPS in the temperature range of 1800° to 3000°F. Although material development is the most crucial factor which determines the feasibility of a reusable radiative TPS, there are many other important aspects associated with the design of a TPS that also have to be examined. The purpose of this memorandum is to identify and discuss some of these important considerations, which include:

1) the choice of TPS,

2) the design concepts of radiative TPS, and

3) the critical problems associated with these designs.

II. Choice of Thermal Protection System[1-11]

Of the various prevailing heat protection concepts, only the ablative system and the radiative system have been considered for application in the preliminary space shuttle studies. Other concepts such as heat sink and transpiration cooled systems are considered to be either unsuitable from the standpoint of weight and complexity or not sufficiently advanced for practical application in the near future. Therefore, a choice would have to be made, most likely, from among the following systems (Figure 1):

1) a refurbishable ablative system,

2) a vehicle design which can accommodate either an ablative or a radiative heat shield, and

3) a radiative system.

It should be noted that discussion is limited to the surface areas subjected to a maximum temperature between 1800° and 3000°F during reentry.

(1) Refurbishable Ablative System

The technology for an ablative heat shield is available, yet its use for the space shuttle may prove to be too costly in a long-range space effort because of the inherent high refurbishment cost and the associated long turn-around time. The concepts of nonreceding porous ablators and other reusable ablators currently under investigation[7] seem promising except that these materials are still in their early stages of development and the prospects of their becoming a reliable system in the immediate future is remote.

(2) Vehicle Design Which Can Accommodate Either an Ablative or a Radiative TPS

Three possible variations in development philosophy can be made within the scope of this concept, according to whether the emphasis is placed on the ablative system or on the radiative system. These variations are as follows:

a. A Structural Design Based on an Ablative TPS Which Can be Retrofitted with a Radiative System When the Latter is Considered Ready for Application.

This approach appears to be reasonable provided that the weight penalty involved in the provision for retrofitting is small. It is considered conservative because emphasis would inevitably be placed on an ablative TPS concept in the course of vehicle development. Thus, no rapid progress in the technology of radiative TPS would be expected in the near future. Under these circumstances, the vehicle developed would most likely be constructed with an ablative TPS, and the prospect of having a radiative system incorporated in a near future would be unlikely.

b. A Structural Design Based on a Radiative TPS Which Can Be Retrofitted With an Ablative System if and When the Former is Found to Be Unreliable or Unsuitable for Reuse.

This approach offers the same degree of safety as the preceding one. If it is followed, the vehicle would be developed on the basis of a radiative TPS. The probability of achieving a reusable radiative TPS is therefore greatly enhanced.

c. A Structural Design Which Can Equally Accommodate Either a Radiative or an Ablative Heat Shield.

This approach places equal emphasis on development of radiative and ablative heat shields. There are two questions associated with this approach, the answers to which exert substantial impact on its practicality:

1) Is it practical to launch parallel development programs for both a radiative and an ablative TPS for the space shuttle within the constraints of budget and time?

2) Is there any significant difference in structural design between this approach and 2a or 2b?

Technologies associated both with the radiative TPS and the ablative TPS can be expected to make reasonable advances in the near future, but the pace of progress for each system will depend to a large extent upon the size of the investment applied in the immediate future. It appears unrealistic to invest heavily and equally in both TPS in association with the shuttle development; sooner or later a choice would have to be made between an ablative and a radiative system to receive concentrated effort.

Therefore, this approach would be practical only if the penalties paid for the provision of equal accommodation of both systems are so small that whichever is selected would have little effect on the vehicle structural weight and the total vehicle cost.

(3) Radiative System

Incorporating a radiative TPS in a space shuttle would represent a major breakthrough both in design and in operation of a reentry vehicle. The radiative TPS has been receiving preferred consideration over an ablative system despite the existing uncertainties regarding the reliability and the reusability of almost all the materials potentially applicable to this system.

If this system concept were adopted, technological advancement would be imperative because it lacks the flexibility of the 2b or 2c approach. The degree of risk involved here is directly related to the heating rate which the surface area is subjected to; the higher the heating rate the higher the risk would be. If the maximum temperature of all the surface areas except the nose and small areas of leading edges could be kept below 2500°F, a temperature within the capability of coated Columbium (Cb) alloys, there would be a relatively small risk involved in the reusability of the heat shield since coated Cb is already in an advanced stage of development. The probability of a catastrophic heat shield failure during a reentry due to coating damage of such a system has been demonstrated to be small[6], and so the uncertainties placed on crew safety and mission success would also be considered to be negligible. Results of preliminary studies on fully reusable, two-stage vehicle systems indicate that an upper temperature limit of 2500°F during reentry on all the surfaces except the nose would be achievable[3,4,5]. Furthermore, if constraints could be placed on the vehicle configuration and the reentry trajectory so that most of the vehicle surfaces could be kept below 2200°F, a temperature within the capability of thoria dispersion strengthened nickel-chromium (TDNiC), the concept of a radiative TPS would be even more attractive.

An early detailed trade-off study among the three versions of approach 2 and the approach 3 would be most useful in identifying the design problems involved in each approach, and in pinpointing the systems which would best provide a stimulus for further technological progress toward an economical reusable reentry vehicle.

III. Design Concepts of Radiative TPS

A variety of combinations of the following subsystems which make up the design of space shuttle TPS with a radiative heat shield can be made:

1) radiation shield and its support,

2) external insulation,

3) internal insulation,

4) load-bearing structure, and

5) active cooling system.

Although a number of combinations are possible, four are preferred (Figure 2):

1) a metallic heat shield with internal insulation,

2) a non-metallic (or composite) external insulation which also acts as a radiator,

3) a metallic heat shield with internal insulation and an external insulation overlay, and

4) a metallic heat shield which is also the primary load-bearing structure.

The first three of these are placed on the primary load-bearing structure, and the last one is an integral heat shield and primary structure.

An active internal cooling system can be incorporated into each of these concepts when the temperature of the internal structures has to be kept at a very low level. Previous studies[3,9] indicate that an active cooling system might provide significant weight advantage over a passive cooling system with insulating layers only.

(1) <u>A Metallic Heat Shield with an Internal Insulation</u>

This concept is most commonly used in current research and development programs[6]. In a typical design, small metallic heat shield panels, 12 in. to 24 in. square and supported by several rigid posts or multiple clips[3] along the panel edges, are used to form vehicle surfaces and to reject by radiation the thermodynamic heat generated during reentry. These panels carry and transmit the local air load only, whereas the main vehicle loadings are supported by an inner structure which is kept at a temperature well below that of the surface either by insulation layers between the heat shield panel and the inner structure, or by a combination of insulation and active cooling. Many metal alloys are potentially applicable to the construction of heat shield panels, but the properties of most of these materials at high temperatures have not been fully characterized.

The advantages offered by this concept are:

1) The metallic heat shield provide a rigid surface.

2) It has potential for many reuses with minimum refurbishment.

3) The metallic heat shield materials are in a more advanced stage of development than non-metallic materials.

The disadvantages of this concept are:

1) The materials applicable to heat shield construction have limited temperature capability.

2) It may be heavier than other designs because coated refractory metals are vulnerable to micrometeoroid impact as well as damages from other foreign materials.

3) Post flight inspection of coated refractory metals would be a problem.

(2) <u>A Non-metallic (or composite) External Insulator</u>

The main feature of this design concept is a light weight, non-receding, rigid material which serves as both a heat radiator and as an insulator. A typical example of an external insulating material would be the brick-like LI-1500[3] developed by Lockheed Missiles and Space Company. This material may be bonded either to a metallic plate to form a heat shield panel or bonded directly to the load-bearing structure.

The advantages of this concept are:

1) Construction of the total thermostructure may be simpler than the preceding concept.

2) It may have a weight advantage[3].

3) It is more tolerant to flaws and damages caused by meteoroids and other foreign materials.

4) It is more tolerant to unexpected high temperature.

The disadvantages of this concept are:

1) Materials of this category are still in an early stage of development.

2) Most of the light-weight insulators have very low strength at room temperature.

3) The temperature of inner structures would continue to rise after the touch-down of a reentry because of the thickness required for insulation.

4) Joining of the insulator to the substrate would be difficult due to the brittleness of these materials.

(3) A Metallic Heat Shield with an Internal Insulation and an External Insulation Overlay.

An external insulation overlay may be applied to a basically metallic heat shield design at locations where the maximum temperature would be very close or possibly exceeding the capability of the metal alloy. The material of this rather thin outer layer could be so chosen that the heat shield is basically non-receding under the expected normal heating rate, but would ablate to absorb heat when subjected to heat pulse beyond the design limit, thus protecting the underlying metallic heat shield. The overlay would also provide extra insulating capability under normal conditions. However, very little experimental work has been done toward the development of this heat shield concept and little is known at this time concerning the availability and selection of a proper material for the outer insulation and the means for attaching the outer layer to the metallic structure.

(4) A Metallic Heat Shield Which also Serves as a Load-bearing Structure.

The concept of a continuous hot structure subjected to both the aerodynamic heating and the primary vehicle loads is attractive for certain vehicle areas such as fins, control surfaces and wing stabs where a relatively high internal temperature is tolerable.

The advantages of this concept are:

1) The structure may have a weight advantage in certain cases.

2) A single wall hot structure would result in a simple construction.

The disadvantages are:

1) It may introduce high thermal stresses and large

deflections wherever a large temperature gradient exists; it is, therefore, unsuitable for areas surrounding an inhabited cabin or anywhere the internal temperature has to be kept at a very low level.

2) A single wall construction is more vulnerable to meteoroid penetration.

3) Repair of a continuous hot structure could be very difficult.

4) Its usefulness is severely limited by the strength loss of materials at high temperatures.

It is felt that, at present state of technology, the hot structure concept could be applied advantageously only for thin structures such as wings and control surfaces where the maximum surface temperature does not exceed 1800°F, which is the upper limit for the conventional super alloys. For very lightly loaded areas, this temperature limit may be raised to 2200°F or even 2500°F in the near future, depending on the progress to be made on TDNiC or coated Cb alloys.

IV Critical Problems

There are many technical problems associated with the design and development of a radiative TPS for which more studies are urgently needed. A list of some of the most important ones include:

1) evaluation and selection of high temperature heat shield and insulation materials,

2) panel flutter,

3) compromise between vehicle performance and structural efficiency,

4) determination of vehicle mass ratio and the allowable weight growth of the structures,

5) potential damage caused by foreign materials,

6) development of a reliable active cooling system,

7) protection of special regions, and

8) fabrication, inspection and repair techniques.

The likelihood of developing a reliable, effective radiative TPS would depend greatly upon the solution of these problems. Details are discussed in the following paragraphs.

(1) Evaluation and Selection of High Temperature Heat Shield and Insulation Materials[6,12]

Several high temperature materials are potentially applicable to the construction of radiative TPS. Yet, all of these materials are still at a research or development stage, and very little engineering data are available from test and hardware applications.

Among the promising metal alloys, TDNiC (with a temperature limit of 2200°F) and coated Cb alloys (with a temperature limit of 2500°F) have been the subjects of a considerable number of experimental studies. A high temperature coating system is yet to be developed for Ta alloys (with a temperature limit of 3600°F).

TDNiC is an oxidation resistant metal which can be used without a protective coating; but its availability, reproducibility, joining techniques, low elongation limit at high temperatures and the low residual strength after long time, high temperature and high stress exposure still represent difficulties not yet overcome. As for Cb alloys, the evaluation and improvement of existing coating systems should be the main object to achieve. LI-1500 (an all silica composite material with a temperature limit of 2500°F) appears promising for external insulation, even though this material has not been fully characterized and its practicality remains to be proven; shrinkage at high temperatures and the difficulties encountered in bonding it to substrate are among the main problems to be solved. It should be pointed out that past experimental evaluations of these materials have been based primarily on static furnace tests of small structural elements; data on high speed hot gas/cold wall reentry simulation tests are not yet available.

For internal insulation, micro-quartz (with a temperature limit of 1800°F) and dyna-flex (with a temperature limit of 2700°F) appear to be the leading candidates. Again, more test data are needed to establish their thermal conductivity, dimensional stability and resistance to acoustic vibration at high temperatures, as well as their compatibility with other materials such as coated refractory metals in a transient heat conduction environment. For insulation at a temperature above 2700°F, zirconia-felt is a material worthy of further investigation.

(2) Panel Flutter

Panel flutter is one of the design problems for a metallic heat shield because of its light-weight, thin-sheet, panel construction. The critical flutter condition is likely to occur during the launch phase at supersonic speed, but may also occur during reentry in the presence of severe aerodynamic heating. Early studies indicated that flutter may be a governing condition in the design of certain types of panel structures.[13] In some cases, large effects on flutter characteristics resulting from small changes in boundary constraints were experienced.

Recent hardware development of radiative metallic heat shields (some of these programs are discussed in reference 6) has emphasized the use of small thin sheet panels, 12 to 24 inch square, with several different supporting conditions to allow for thermal expansion and to facilitate refurbishment. In most of these development programs, however, panel flutter has not been included as a part of the evaluation of structures and materials.

(3) Compromise between the Vehicle Aerodynamic Performance and the Structural Efficiency.

The aerodynamic heating to which a spacecraft would be subjected during reentry is primarily a function of vehicle configuration and the reentry trajectory; the design of a vehicle TPS is extremely sensitive to the extent of this aerodynamic heating. In general, the difficulties involved increase with an increase of maximum surface temperatures of the vehicle.

A well balanced design should take into consideration the tradeoff between aerodynamic performance and the structural efficiency. It is believed that a careful study of these considerations early in the space shuttle program could lead to a significant simplification of the TPS and a great reduction of the technological development needed for the structural design. From the viewpoint of TPS design, a reasonable goal would be to define the aerodynamic contour and the reentry flight path of the vehicle so that the external surface, except possibly a small area of the nose section and some leading edges, would not be subjected to temperatures greater than 2200°F. Thus, an uncoated metallic heat shield (with TDNiC in mind) could be made that would ensure a reliable and reusable vehicle with simple operation and maintenance procedures. The small but critical surfaces of the nose and leading edges could be cooled actively or use a replaceable ablative material.

(4) <u>Determination of Vehicle Mass Ratio and the Allowable Weight Growth of the Structures.</u>

The mass ratio (ratio of structural weight to total stage weight), and hence the weight sensitivity, of the fully reusable two-stage space shuttle will be more critical than that of current space launch vehicles. Preliminary weight estimates[3,4,5] indicate that the payload fraction of a space shuttle is very low in comparison with earlier vehicles. Consequently, weight control of both the TPS and the load-bearing structures is extremely important. Since there are more uncertainties associated with the design of TPS than that of the primary load-bearing structures, weight control of the former appears to be more difficult.

Past records of aerospace vehicle developments show that weight growth always occurs. Since the payload margin of a space shuttle is very small to begin with, it is desirable to establish a guideline to limit the mass ratio of each vehicle stage. A detailed weight analysis, including an estimate of allowable weight growth and an assessment of the factors which could lead to such a growth, should be made early in the shuttle program for promising design concepts. Possible ways to reduce the weight should also be investigated.

For example, total weight reduction may be achieved through replacing some of the metallic inner structures with composite materials.[14] Estimates should be made of the potential weight savings of such a replacement or redesign and the probability of keeping the vehicle from prohibitive weight growth.

(5) <u>Potential Damages Caused by Foreign Materials</u>

The meteoroid environment in space may present significant hazards to the heat shield of a spacecraft, particularly if coated refractory metals are used. Based on an exposed heat shield area of 3000 ft.^{2}, a mission duration of 3 days and an average altitude of 210 NM, a Bellcomm study[15] of the effects of meteoroid impact on a columbium heat shield coated by the Sylvania R512E coating system shows the following results:

1) The number of punctures through the protective coating, for a probability of 0.999 and with a meteoroid environment uncertainty factor of 2, could be as high as 100, with a total damaged area of no more than 22.0×10^{-3} in.^{2} These punctures would expose the Cb substrate to the surrounding oxygen during reentry, which may lead to excessive oxidation of the material and, hence, to create problems for post flight inspections and refurbishment.

2) The required heat shield thickness for zero punctures of the heat shield would be approximately 0.14 in. for a probability of 0.999 and 0.06 in. for a probability of 0.99, as compared to the current estimate of 0.012 to 0.025 in.

These results indicate that meteoroid impact could be a decisive design criterion for the TPS of the space shuttle. It appears advisable that this factor be taken into consideration in the conceptual studies as well as in the early stages of hardware development.

Other factors, such as rock impacts from the runway during landing and rain erosions on the heated structures during reentry may also cause damages to the TPS.

(6) Development of a Reliable Active Cooling System[8,9,11]

A choice of either a passive (insulation layers only) or an active (insulated and cooled by a circulating coolant) cooling system can be incorporated into a radiative TPS.

A passive system would be reliable and simple in construction; yet it requires a large thickness of insulation which might limit its applications. An active cooling system may have a weight advantage in many cases and its weight would not be a strong function of the design surface temperature and heating time. It also requires less insulation thickness, and is particularly suitable for areas where efficient use of interior space is important. However, on the basis of current technology, this system would be more costly and less reliable than a passive system. The development of a simple and reliable active cooling system would be a technological advancement which could greatly facilitate the application of a radiative TPS.

(7) Protection of Special Regions

To provide adequate thermal protection, structural integrity and compatibility to the total vehicle, consideration must be given in the design of the TPS to regions such as cavities, penetrations, and interfaces between dissimilar materials as well as to structures other than the heat-shield surfaces. These special regions which present unique design problems include:

1) locking and sealing of hatches, windows, landing gear doors, and the structural attachments between the booster and the orbiter vehicles;

2) structural interface between hot structures (such

as a wing) and cool structures (such as the load-bearing structure of the fuselage);

3) junctions between ablative heat shields and radiative heat shields as may be needed in the area of the nose and leading edges; and

4) turbojet landing engines.

(8) Fabrication, Inspection and Repair Techniques

Some fabrication difficulties exist in all of the design concepts discussed in this memorandum, particularly in fastening and joining the heat shield material or adhering it to the substrate. Post flight inspection and damage repair techniques are also an important part of the vehicle operations. A simple, reliable non-destructive testing (NDT) method for field inspection of the coated refractory metals is needed. In order to shorten the turn-around time the possibility of a built-in, automatic NDT System for the vehicle structure, including the TPS, should be investigated. Several coating repair techniques have been developed, but the reliability and reusability of the repaired coatings must be improved.

V. Concluding Remarks

Radiative and ablative TPS have been evaluated for the Space Shuttle vehicle application. In order to provide a stimulus for further technological progress, and at the same time to reasonably assure the development of a successful, reusable shuttle system, a vehicle design which can accommodate either an ablative or a radiative heat shield appears to be a good choice.

The advantages and disadvantages of four versions of radiative TPS design have been discussed. Due to uncertainties on the reliability and the reusability of high temperature materials as well as the maximum temperatures of the vehicle surfaces, a definite preference can not be made at this time. It should be noted, however, that a TPS with a metallic heat shield and internal insulation appears to be in a more advanced stage of development than other radiative TPS.

A number of problems associated with the design of a radiative TPS for the space shuttle have been identified, and the need of future research and development have been discussed. The most critical problems are:

1) evaluation and selection of heat shield and insulation materials,

2) control of the structural weight,

3) control of the maximum surface temperature,

4) the effects of meteoroid impact, and

5) design of special vehicle areas such as the locking and sealing of landing gear doors and the structural interface between the carrier and the orbiter stages.

It should be emphasized that the TPS should not be treated as an isolated system. In selecting its design, the vehicle configuration should be examined; the aerodynamic performance should be traded with structural effectiveness; and the total structural system of the vehicle must be synthesized accordingly. It is only through such an integrated approach that a most efficient structural system can be designed.

C. C. Ong

1013-CCO-baw C. C. Ong

Attachment

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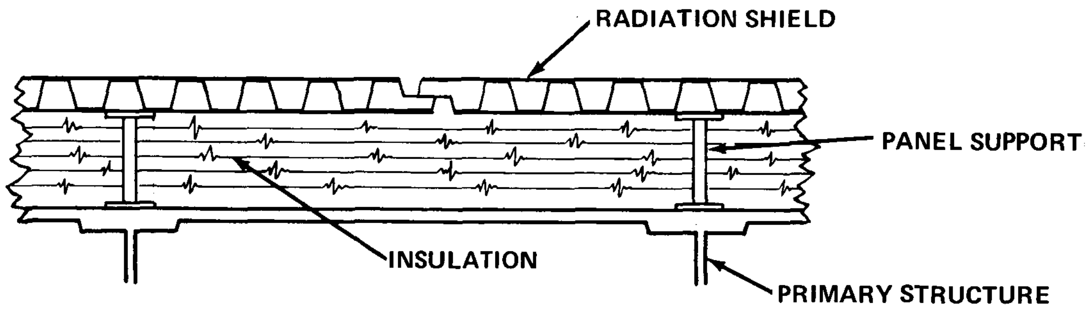
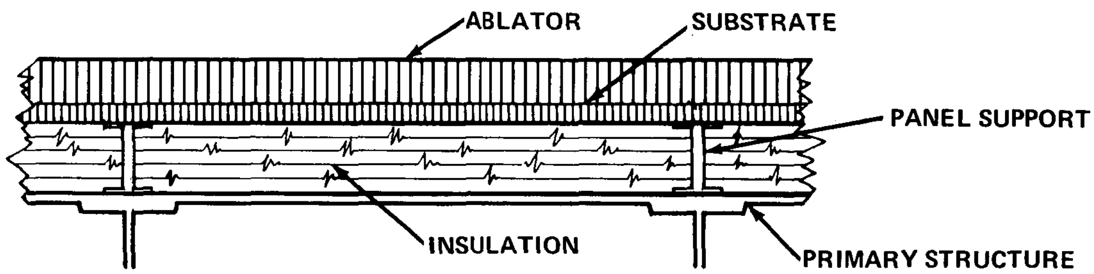
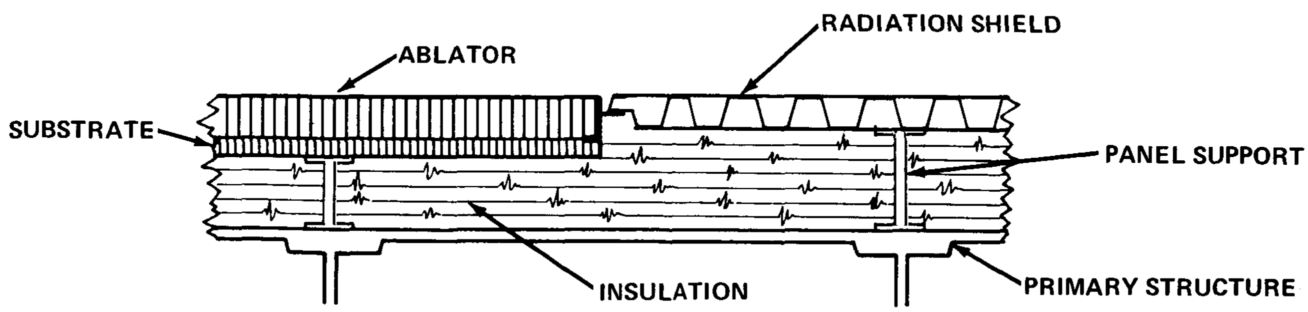


(A) RADIATIVE SYSTEM

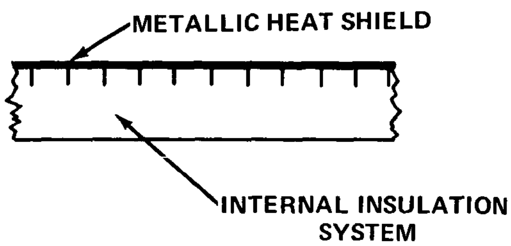


(B) ABLATIVE SYSTEM

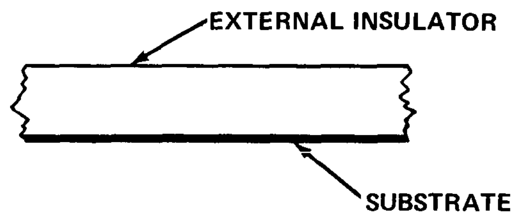


(C) INTERCHANGEABLE SYSTEM

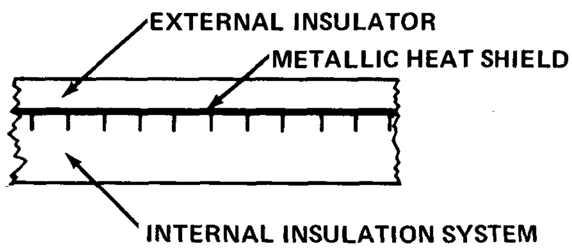
FIGURE 1 - THERMAL PROTECTION SYSTEM CONCEPTS



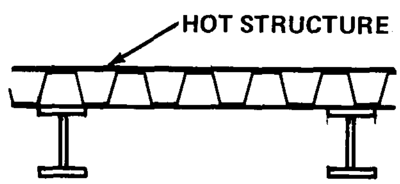
(A) METALLIC HEAT SHIELD/INTERNAL INSULATION



(B) NON-METALLIC EXTERNAL INSULATION



(C) EXTERNAL INSULATION/METALLIC HEAT SHIELD/INTERNAL INSULATION



(D) INTEGRATED HEAT SHIELD AND LOAD-BEARING STRUCTURE

FIGURE 2 - RADIATIVE THERMAL PROTECTION SYSTEMS

BELLCOMM, INC.

Subject: Radiative Thermal Protection System Considerations for the Space Shuttle

From: C. C. Ong

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